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PRELIMINARY STUDY OF ADVANCED HYPERSONIC RESEARCH AIRCRAFT

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16. Abstract

In this study some important flight constraints for hypersonic research aircraft are reviewed. Maximum lift-drag ratios for state-of-the-art hypersonic aircraft in trimmed flight have been derived from wind-tunnel tests. Weights, cruise times, and ranges for rocket-boosted research airplanes with air-breathing cruise engines are estimated and compared. Both horizontal take-off from the ground and air launch from a subsonic airplane are considered.

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SYMBOLS

$\bar{c}$	mean aerodynamic chord, ft
c_p	specific heat at constant pressure, Btu/lb °R
D	drag, lb
g	gravitational acceleration, ft/sec^2
h	altitude, ft
h	enthalpy, Btu/lb
I	specific impulse, lb sec/lb
l	body length, ft
L	lift, lb
m	mass, slug
M	Mach number
p	pressure, atm
q	dynamic pressure, $\frac{1}{2}\rho V^2$, psf
$\dot{q}_c$	convective heat-transfer rate, Btu/ft^2 sec
$\dot{q}_r$	radiative heat-transfer rate, Btu/ft^2
r	distance from earth center, ft
R	nose radius, in.
R	range, ft
Re/ft	Reynolds number based on length of one foot
$R_{\bar{c}}$	Reynolds number based on mean aerodynamic chord
R_l	Reynolds number based on body length
t	time, sec
T	thrust, lb

T	temperature, °R or °K as specified
V	velocity, fps
W	weight, lb
γ	flight-path angle, deg
ϵ	emissivity
Λ	wing sweepback angle, deg
ρ	density, slugs/ft^3
$(\bar{\ })$	average

Subscripts

LE	leading edge
max	maximum
n,n+1	successive time points
o	reference
re	radiation equilibrium
stag	stagnation
t	total
w	wall

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SUMMARY

In this study some important flight constraints for hypersonic research aircraft with air-breathing cruise engines are reviewed; maximum lift-drag ratios for hypersonic aircraft in trimmed flight are derived from wind-tunnel tests; and weights, cruise times, and ranges are estimated and compared for rocket-boosted research airplanes that take off horizontally from the ground or are air launched from a subsonic airplane. For flight at Mach numbers greater than about 6 to 8, materials that will withstand temperatures above 2,800° R must be developed for the vehicle nose and other regions of near stagnation heating. Maximum lift-drag ratios for state-of-the-art hypersonic cruise vehicles in trimmed flight range from about 3.6 at Mach number 6 to 3.1 at Mach number 12. It appears that a rocket-boosted research airplane weighing about 100,000 pounds at take-off and capable of cruising at a Mach number of 12 for 5 minutes is feasible and merits detailed investigation. The indicated gross weight of 65,000 pounds for an air-launched airplane would rule out air launch from a B-52 bomber, but launch from a C-5A airplane appears feasible.

INTRODUCTION

To advance the state of the art of hypersonic technology so that successful hypersonic aircraft can be developed, a hypersonic research airplane might be used to aid and support ground-based research. For economic reasons, such an airplane should be as small and simple as possible, and its performance should not depend upon large advances in the state of the art. If an existing and proven rocket engine were used as the primary propulsion system, at least initially in the development of the research vehicle, aerodynamic, structural, and flight operations research could be performed prior to the development of a hypersonic air-breathing propulsion system. The aircraft might then be used as a test bed during the development of an air-breathing propulsion system.

At the Ames Research Center, a brief preliminary study has been made of the airplane and fuel weights required for cruise flights as long as 8 minutes at Mach numbers of 6, 8, 10, and 12. Both rocket-boosted horizontal take-off from the ground and air launch from a B-52 bomber at 43,000 feet have been studied. A possible flight domain over which the aircraft would operate has been established, and various flight domain constraints have been noted. Because the aircraft weight depends on the cruise lift-drag ratio, maximum lift-drag ratios for hypersonic aircraft have been reviewed and reasonable values selected. The selections were based on experimental results (from ground-based facilities) for typical hypersonic cruise configurations in trimmed flight.

Results presented pertain to (1) flight domain constraints for hypersonic aircraft; (2) maximum lift-drag ratios for typical hypersonic aircraft; and (3) approximate weights, cruise times, and ranges for rocket-boosted research airplanes with air-breathing engines.

FLIGHT DOMAIN CONSTRAINTS FOR HYPERSONIC AIRCRAFT

Before looking in detail at a hypersonic research aircraft, it is worthwhile to examine the environmental constraints that influence the choice of a trajectory. A typical climb trajectory and several flight constraints are shown in figure 1 on a plot of altitude versus velocity. The climb trajectory (dashed curve) is typical of a hypersonic cruise vehicle and was determined by constraining the sonic boom to not more than 3 psf, the dynamic pressure to not more than 1800 psf, and the inlet duct pressure to not more than 150 psia. It should be emphasized that this path is preliminary and is shown only for illustration. The sonic boom constraint probably could be ignored for a research vehicle. A rough maximum limit for the dynamic pressure constraint is about 2000 to 2800 psf (see, e.g., ref. 1). A duct pressure constraint of about 130 to 300 psia is likely (ref. 1), but further investigation is needed in this area.

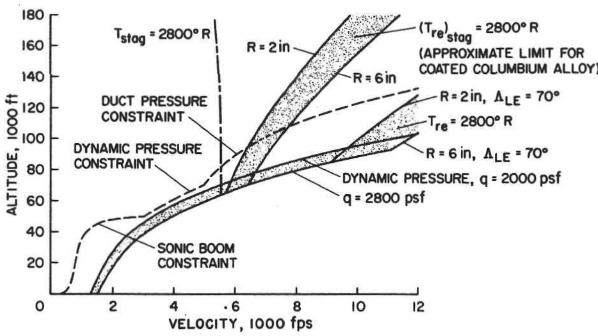


Figure 1.- Some representative flight domain constraints.

Some temperature constraints are also indicated in figure 1. An operational limit of about 2800° R is indicated for an advanced, coated columbium alloy (see, e.g., ref. 2). An isotherm is shown for a stagnation temperature of 2800° R, but at the stagnation region of an aircraft, the constraint is not this severe because of the effect of radiation cooling. Also shown on the plot is the flight region in which the radiation equilibrium temperature at the nose stagnation point of a vehicle is 2800° R. This region has nose radius bounds of 2 and 6 inches. The procedure used to compute this zone is derived in appendix A. For an aircraft to operate beyond this region, materials with higher temperature capability will be needed or a regenerative cooling or ablation system must be used for the vehicle nose. For highly swept wings, the cooling problem will be considerably alleviated, as shown for a 70° swept wing. The isotherm lines for this wing with leading-edge radii of 2 and 6 inches were estimated using Eckert's reference enthalpy technique for a turbulent boundary layer (ref. 3).

MAXIMUM LIFT–DRAG RATIOS FOR HYPERSONIC AIRCRAFT

Lift-drag ratios used in this preliminary analysis of hypersonic research aircraft were obtained from wind-tunnel tests of the hypersonic models illustrated in figure 2. The results of the tests, which are discussed in detail in reference 4, have been used as the basis for the flight extrapolations

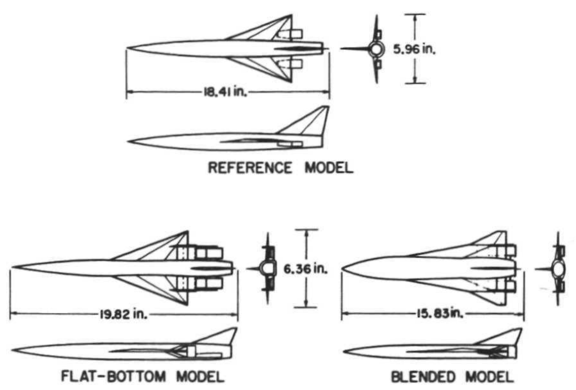


Figure 2.- Test configurations.

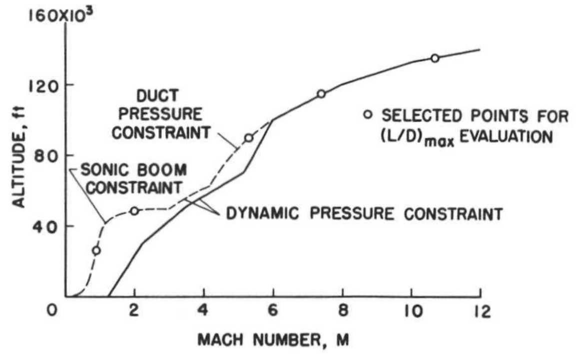


Figure 3.- Climb profile.

herein. The results indicate that each model developed about the same maximum L/D; hence, a single set of values (table 1) that essentially applies to all three configurations has been used in this study.

Reynolds numbers for correcting the lift-drag ratios from the wind tunnel to flight conditions were obtained from the assumed flight profile shown with the indicated points in figure 3. The points identify some of the Mach numbers at which data were obtained in wind tunnels at Ames, and these points were selected for the $(L/D)_{max}$ evaluations. Two possible profiles are shown for $M < 6$. The dashed line indicates the effect of a sonic boom constraint of about $\Delta p \leqslant 3$ psf; for cruise flight at $M < 6$ this profile, or a similar one, might be used. However, for a hypersonic research airplane that climbs over uninhabitated ground, the sonic boom constraint might be omitted, and a climb profile such as indicated by the solid line could be used. For the solid line profile, the velocity increases to about $M = 1.2$ ($V \approx 1200$ fps) near zero altitude. Then a dynamic pressure contour line of about 2000 psf is followed to about $M = 5.2$ ($V \approx 5000$ fps). The contour from $M = 5.2$ to $M = 6$ follows a typical inlet duct pressure constraint of about 150 psia, although the research aircraft could continue to follow the dynamic pressure constraint if desired. It is envisioned that a research vehicle would cruise at $M = 6$ at an altitude of about 100,000 ft, or for higher cruise Mach numbers, it would continue to climb to higher altitudes along the solid curve and then cruise at a designated Mach number somewhere between $M = 6$ and 12.

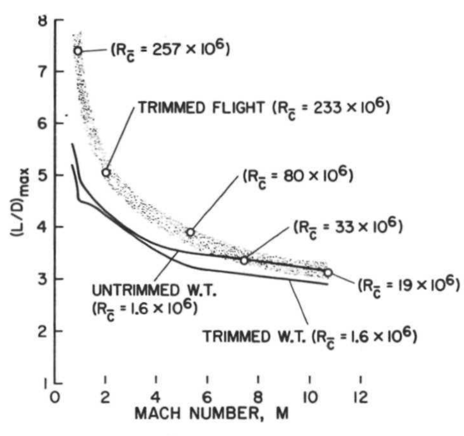


Figure 4.- $(L/D)_{max}$ vs. M for test configurations with all-turbulent boundary layers.

To extrapolate the wind-tunnel results to the flight conditions, estimated skin-friction drag coefficients for the tunnel conditions were subtracted from the experimental drag coefficients, and turbulent skin-friction values for the flight conditions were added. The Eckert reference temperature methods were used for the computations (refs. 3 and 5).

Values of $(L/D)_{max}$ versus Mach number are shown in figure 4 for all-turbulent boundary layers. (At hypersonic speeds in the wind-tunnel tests the boundary layers were laminar, but the data in figure 4 have been corrected to all-turbulent boundary layers.) The upper solid curve is for the untrimmed configurations in two Ames wind tunnels (see table 1) at a Reynolds number of about 1.6×10^6, based on

mean aerodynamic chord. The lower solid curve is for the same vehicles trimmed with positive directional stability at $(L/D)_{max}$ for wind-tunnel conditions. The circles indicate the values of $(L/D)_{max}$ for full-scale trimmed flight with positive directional stability. The Reynolds numbers shown for the flight conditions were determined from the flight profile in figure 3 for a cruise aircraft 285 ft long with a mean aerodynamic chord of 87.4 ft. For a hypersonic research airplane, the Reynolds numbers probably would be about one third of those shown in figure 4 because of the expected smaller size of the research airplane (maybe 100 ft long). From computations it can be shown, however, that the reduction in $(L/D)_{max}$ would be negligible in the hypersonic Mach number range from about 6 to 12.

It is clearly illustrated in figure 4 that at hypersonic speeds, $(L/D)_{max}$ decreases about 0.4 in the wind tunnel when the vehicle is longitudinally trimmed and positive directional stability is provided. With continued research this increment probably can be diminished somewhat by the use of more efficient methods to obtain stability and trim.

The lift-drag ratios (fig. 4) are higher in flight than in the wind tunnel because the Reynolds numbers are much greater in flight, and the turbulent skin-friction drag coefficient is decreased. However, at hypersonic Mach numbers the flight lift-drag ratios are only about 0.3 to 0.5 higher than those obtained in the wind tunnels. For example, at a Mach number of about 10.7 the trimmed flight L/D is about 3.2 as compared with about 2.9 in the wind tunnel (see also table 1).

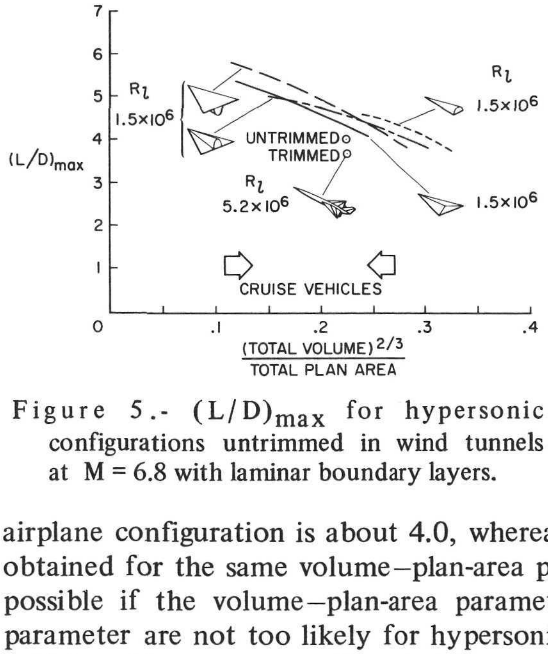
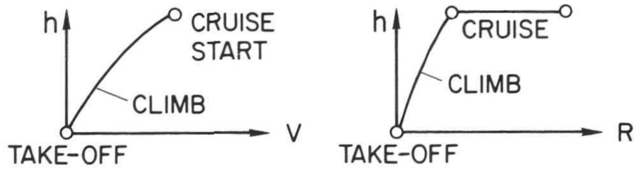


Figure 5.- $(L/D)_{max}$ for hypersonic configurations untrimmed in wind tunnels at M = 6.8 with laminar boundary layers.

Higher values of $(L/D)_{max}$ might be obtained at hypersonic Mach numbers by optimizing the vehicle shapes. This possibility is suggested by the data plotted in figure 5 which shows $(L/D)_{max}$ as a function of the volume–plan-area parameter. (Note that the data on this figure are for laminar boundary layers, and the Reynolds numbers are based on body length rather than on mean aerodynamic chord.) The results shown are from the Langley Research Center (refs. 6 and 7) for various high lift-drag ratio shapes tested at M = 6.8. The untrimmed and trimmed values of $(L/D)_{max}$ for the complete aircraft configuration tested at Ames are compared with these results. At M = 6.8, $(L/D)_{max}$ for the untrimmed airplane configuration is about 4.0, whereas for the simpler shapes, values of about 4.5 have been obtained for the same volume–plan-area parameter. It also appears that some gain in $(L/D)_{max}$ is possible if the volume–plan-area parameter can be diminished, but significant decreases in this parameter are not too likely for hypersonic research aircraft because of scaling effects. For cruise vehicles, this parameter will probably range from about 0.14 to 0.24.

AIRCRAFT WEIGHTS, CRUISE TIMES, AND RANGES

Since this was a preliminary study of hypersonic research aircraft, only the climb and cruise phases of the flight profile (as sketched) were considered. Unpowered glide flight, envisioned for the landing phase, was not studied.



Formulas for the Climb Phase

For the climb phase the earth was assumed to be flat and the thrust to be alined with the velocity vector. Under these assumptions, the weight fraction for climb (derived in appendix B) is expressed as

$$\frac{dW}{W} = -\frac{dV + \frac{g}{V}\,dh}{gI\left(1 - \frac{D}{T}\right)} = -\frac{dV + g\,\frac{dh}{dV}\,\frac{dV}{V}}{gI\left(1 - \frac{D}{T}\right)} \tag{1}$$

In all computations, incremental steps along the flight path were assumed. For a step from point n to point $n+1$, $dh/dV \approx \Delta h/\Delta V$, and

$$\ln \frac{W_{n+1}}{W_n} = -\frac{1}{g\bar{I}}\left(\frac{V_{n+1} - V_n + g\,\frac{h_{n+1} - h_n}{V_{n+1} - V_n}\,\ln\frac{V_{n+1}}{V_n}}{1 - \frac{\bar{D}}{\bar{T}}}\right) \tag{2}$$

where the bar over a value indicates the average of the values at points n and $n+1$. Incremental time and range are given by

$$\Delta t = -\frac{W_n - W_{n+1}}{\dot{W}} = -\frac{W_1}{\dot{W}}\left(\frac{W_n}{W_1} - \frac{W_{n+1}}{W_1}\right) \tag{3}$$

and

$$\Delta R = \frac{1}{2}\left(V_n + V_{n+1}\right)\left(t_{n+1} - t_n\right) \tag{4}$$

where w_1 is the take-off weight, and $\dot{w} = -T/I$, with thrust T and specific impulse I assumed constant.

Formulas for the Cruise Phase

For the cruise phase the weight fraction is given by the Breguet equation written so that the cruise time can be specified as an input. For an $(L/D)_{max}$ cruise at constant velocity,

$$\ln \frac{W_{n+1}}{W_n} = - \frac{\Delta t \left(1 - \frac{V^2}{gr}\right)}{I \left(\frac{L}{D}\right)_{max}} \tag{5}$$

Incremental range is given by

$$\Delta R = V \ \Delta t \tag{6}$$

Computed Weights, Cruise Times, and Ranges for Research Airplanes

Weights, cruise times, and ranges have been computed for rocket-boosted research airplanes that take off horizontally from the ground or are air launched from a B-52. For the cruise phase, advanced air-breathing engines were assumed.

In figure 6 the assumed flight profiles are compared for the ground take-off and the air launch conditions. For the ground take-off, the velocity increases to about 1200 fps near zero altitude. The aircraft then follows a dynamic pressure profile ($q \approx 2000$ psf) to a velocity of about 5000 fps. The remainder of the profile path is that defined previously in figure 3. Possible cruise Mach numbers of 6, 8, 10, and 12, as indicated in figure 6, were considered. For the air launch, an initial velocity of 760 fps and an altitude of 43,000 feet are assumed in accordance with B-52 flight capability. The flight path follows the dashed line to the point at which it intersects the ground take-off path, then is identical to the ground take-off path. In the computation of the weight fractions and ranges, five straight-line increments (see fig. 6) were assumed to the point representing a cruise Mach number of 6. For the cruise Mach numbers above 6, the straight-line increments shown on the plot above M = 6 were used. Experience has shown that relatively long increments can be used with little loss in accuracy.

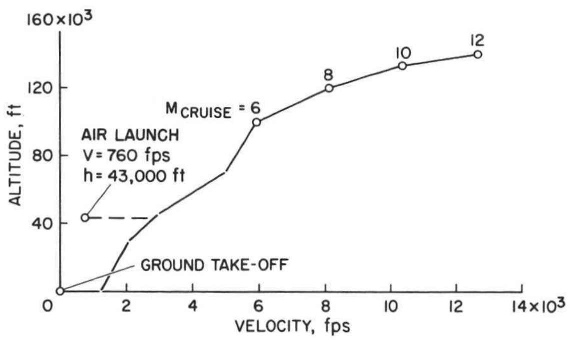


Figure 6.- Flight profiles studies.

Assumed values used in the calculations made in the present study are given in table 2. A J-2 rocket engine using liquid hydrogen and oxygen was specified as the primary propulsion system for the climb phase of the trajectory. The thrust capability was assumed to be 2×10^5 pounds and the

average drag $\bar{D}$ during climb to be 5 percent of the average thrust $\bar{T}$.[1] The values of L/D for the cruise phase were determined from figure 4. For the cruise phase either a ramjet or scramjet was envisioned. Take-off weights of 50,000, 100,000 and 150,000 pounds were specified, and cruise times of 3, 5, and 8 minutes were considered.

Weights at the end of cruise, computed as a function of cruise Mach number, are shown for ground take-off in figure 7 and are listed in table 3. For the take-off weights and cruise times assumed, the weights at end of cruise are much lower for a cruise Mach number of 12 than for, say, a cruise Mach number of 6 or 8. These weights can be considered to represent the allowable unfueled aircraft weights. Because of the high impulse of the air-breathing engines, most of the fuel used is required for the climb phase, and the changes in cruise time have little effect on the vehicle weights. Results from this study indicate that at a Mach number of 6 about 55 percent of the take-off weight would be airframe and engine (plus payload) and about 45 percent would be fuel. At M = 12, however, 68 percent would be fuel.

The fuel used is plotted in figure 8 as a function of cruise Mach number for a cruise time of 5 minutes. It is seen that the fuel used as a percentage of the gross take-off weight increases considerably with increase in cruise Mach number for both ground take-off and air launch. The differences between the percentages of fuel used for ground take-off and for air launch are small, ranging from about 7 percent at M = 6 to about 4 percent at M = 12. The actual differences in fuel

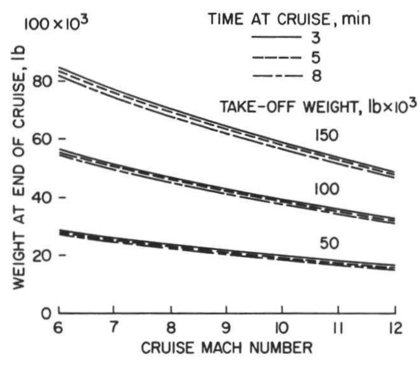


Figure 7.- Weight at end of cruise vs. cruise Mach number.

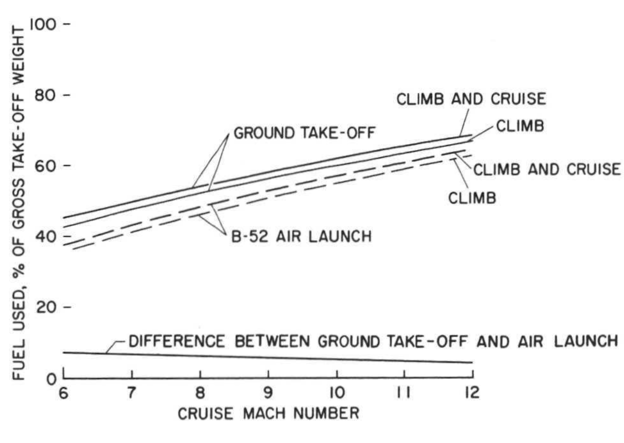


Figure 8.- Fuel used vs. cruise Mach number for cruise time of 5 minutes.

[1] For rocket-boosted aircraft, which accelerate rapidly, constant values of $\bar{D}/\bar{T}$ from 0 to 10 percent can be assumed without appreciable change in the computed weight at end of climb. For $0 \leqslant \bar{D}/\bar{T} \leqslant 0.1$ during climb to cruise altitude, the percent spread in weight at end of climb (begin of cruise) is as follows:

Cruise M	Percent weight spread
6	6.0
8	7.7
10	9.8
12	11.2

weights can be considerable, however, if the gross take-off weights are greatly different. The plots also demonstrate that practically all of the fuel is used during the climb phase for both the ground take-off and the air launch cases. For example, with ground take-off and cruise at M = 12, 66 percent of the gross take-off weight is used for climb and only about 2 percent additional for cruise.

The range (climb plus cruise) versus the cruise Mach number is plotted in figure 9 for an airplane take-off weight of 100,000 pounds and a cruise time of 5 minutes. Note that for a Mach number of 12 the range is over 700 nautical miles for the climb plus cruise phases. It is likely that the range would be at least doubled if the landing phase were included.

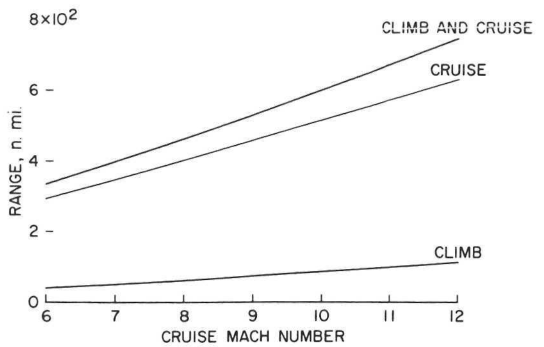


Figure 9.- Range vs. cruise Mach number.

Research aircraft weight estimates for cruise at M = 12 for 5 minutes are listed in table 4. Vehicle weights for air launch and ground take-off concepts are compared. For the McDonnell air-launched test aircraft concept, an RL-20 high pressure rocket was assumed for boost and a scramjet for 5 minutes cruise. The gross weight is estimated to be about 65,000 pounds. For the ground launched vehicle, a gross weight near 100,000 pounds appears to be necessary. The gross weight of 65,000 pounds for the air launch is too high for a B-52 bomber. It is, however, well within the capability of a C-5A airplane.

Although these weight estimates are preliminary, the agreement of the percentages of the empty and gross weights with the results given in figure 8 is very good. For example, once again we note that for a ground take-off and 5-minute cruise flight at M = 12 about 32 percent of the gross weight is airplane weight and about 68 percent is fuel weight. For air launch about 35 percent is airplane weight and about 65 percent fuel weight.

CONCLUSIONS

In this preliminary study of hypersonic research aircraft some important flight constraints have been reviewed; maximum lift-drag ratios for hypersonic aircraft in trimmed flight have been derived from wind-tunnel tests; and weights, cruise times, and ranges have been estimated and compared for rocket-boosted research airplanes which take off horizontally from the ground or are air launched from a subsonic airplane. Some of the most pertinent conclusions are as follows:

1. For flight at Mach numbers greater than about 6 to 8, materials must be developed that will withstand temperatures greater than 2800° R at the vehicle nose and other regions of near stagnation heating. In lieu of such materials, regenerative cooling or ablation systems will be required.

2. Maximum lift-drag ratios for state-of-the-art hypersonic cruise vehicles in trimmed flight will probably range from about 3.6 at Mach number 6 to about 3.1 at Mach number 12.

3. A rocket-boosted research airplane weighing about 100,000 pounds at take-off and capable of cruising at Mach number 12 for 5 minutes appears feasible and merits detailed investigation. The indicated gross weight of 65,000 pounds for a similar air-launched airplane would rule out air launch from a B-52 bomber but not from a C-5A airplane.

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APPENDIX A

DERIVATION OF STAGNATION–POINT RADIATION EQUILIBRIUM TEMPERATURE

In this section an expression is derived for the radiation equilibrium temperature in terms of the free-stream velocity and density. A procedure is given by which isotherms can be included on a plot of altitude versus velocity (see fig. 1).

For convective heating a simplified form (ref. 8) of Fay and Riddell's heating-rate relationship (ref. 9) is

$$\sqrt{R}\,\dot{q}_c = 0.42\sqrt{p_{stag}}\,(h_t - h_w) \tag{A1}$$

where $\dot{q}_c$ is in Btu/ft^2 sec; the nose radius R is in ft; the stagnation pressure p_{stag} is in atmospheres; and the total enthalpy h_t and wall enthalpy h_w are in Btu/lb.

The radiative-heating rate is given by the well known Stefan-Boltzmann relationship:

$$\dot{q}_r = \epsilon\sigma T_w{}^4 \tag{A2}$$

Here the wall temperature T_w is in °K. The emissivity ϵ is assumed to be unity, and the Stefan-Boltzmann constant (ref. 10) is

$$\sigma = 5.6697\times10^{-2}\ \mathrm{W/cm^2\ {}^\circ K^4}$$

When equation (A2) is converted to the same units used in equation (A1),

$$\dot{q}_r = 0.4759\left(\frac{T_w}{1000}\right)^4 \tag{A3}$$

where $\dot{q}_r$ is in Btu/ft^2 and T_w is in °R.

From the Newtonian approximation, $p_{stag} = \rho V^2$, the stagnation pressure in atmospheres can be expressed in terms of the free-stream density ρ in slugs/ft^3 and free-stream velocity V in fps as

$$p_{stag} = 1.125\left(\frac{\rho}{\rho_o}\right)\left(\frac{V}{1000}\right)^2 \tag{A4}$$

where the reference sea level density is $\rho_o = 0.00238$ slug/ft^3.

The enthalpy difference, $h_t - h_w$, is given as

$$h_t - h_w = h_t - h - (h_w - h)$$

$$\approx 20\left(\frac{V}{1000}\right)^2 - c_p(T_w - T) \tag{A5}$$

For $c_p = 0.24$ Btu/lb °R and $T_w >> T$,

$$h_t - h_w \approx 20\left(\frac{V}{1000}\right)^2 - 240\left(\frac{T_w}{1000}\right)^2 \tag{A6}$$

Now for $\dot{q}_c = \dot{q}_r$ and $T_{re} = T_w$, equations (A1), (A3), (A4), and (A6) combine to give

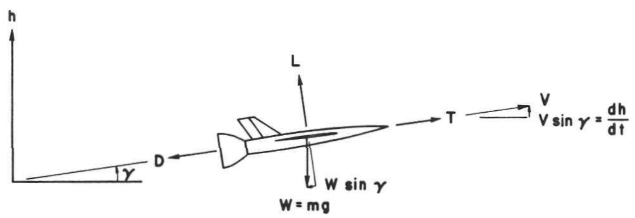
$$\frac{1}{\sqrt{R}}\sqrt{\frac{\rho}{\rho_o}} = \frac{10.69\left(\frac{T_{re}}{1000}\right)^4}{\left(\frac{V}{1000}\right)\left[20\left(\frac{V}{1000}\right)^2 - 240\left(\frac{T_{re}}{1000}\right)\right]} \tag{A7}$$

The procedure for generating temperature T_{re} contours on a plot of altitude versus velocity is as follows:

1. Specify values of T_{re} in °R.
2. For each T_{re} specify values of V in fps.
3. From equation (A7) solve for $(1/\sqrt{R})\sqrt{\rho/\rho_o}$.
4. For values of R in feet, solve for ρ/ρ_o.
5. With the computed values of ρ/ρ_o for input, look up values of altitude in feet from the tables in reference 11.

APPENDIX B

DERIVATION OF WEIGHT FRACTION FORMULA FOR CLIMB PHASE OF FLIGHT PROFILES



For the climb phase it is assumed that the earth is flat and the thrust is alined with the velocity vector. Under these assumptions the sum of the forces in the velocity direction gives

$$T - D - W \sin \gamma = m \frac{dV}{dt} \tag{B1}$$

where

$$\sin \gamma = \frac{1}{V} \frac{dh}{dt} \tag{B2}$$

With equation (B2) substituted into equation (B1),

$$\left(1 - \frac{D}{T}\right) T \frac{g}{W} dt = dV + g \frac{dh}{V} \tag{B3}$$

The thrust is expressed as

$$T = -\dot{w}I = - \frac{dW}{dt} I \tag{B4}$$

With equation (B4) substituted into equation (B3),

$$\frac{dW}{W} = - \frac{dV + \frac{g}{V} dh}{gI\left(1 - \frac{D}{T}\right)} = - \frac{dV + g \left(\frac{dh}{dV}\right)\left(\frac{dV}{V}\right)}{gI\left(1 - \frac{D}{T}\right)} \tag{B5}$$

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TABLE 1.- VALUES OF $(L/D)_{max}$ FOR CONFIGURATIONS IN FIGURE 2

	Wind tunnel (corrected to all-turbulent boundary layer)					Flight			
M	Ames wind-tunnel facility	$Re/ft \times 10^{-6}$	$R_{\bar{c}} \times 10^{-6}$	Untrimmed $(L/D)_{max}$	Trimmed $(L/D)_{max}$	Altitude, ft	$Re/ft \times 10^{-6}$	$R_{\bar{c}} \times 10^{-6}$	Trimmed $(L/D)_{max}$
0.9	6×6-ft transonic	3.5	1.6	5.04	4.62	26,000	2.94	257.0	7.40
2.0	6×6-ft transonic	2.5	1.1	4.30	4.21	48,500	2.55	222.0	5.06
5.3	3.5-ft hypersonic	3.5	1.6	3.50	3.22	90,000	0.91	79.5	3.91
7.4	3.5-ft hypersonic	3.5	1.6	3.37	3.09	115,000	0.38	33.2	3.34
10.7	3.5-ft hypersonic	2.0	0.9	3.15	2.89	135,000	0.22	19.2	3.15
	$l = 1.48$ ft $\bar{c} = 0.46$ ft					$l = 285$ ft $\bar{c} = 87.4$ ft			

TABLE 2.- ASSUMED INPUT VALUES

During climb:

Saturn J-2 LH_2/LO_2
$I = 420$ sec = constant
$T = 2 \times 10^5$ lb = constant

B-52 launch point:

$V = 760$ fps, $h = 43,000$ ft

During cruise:

M	h, ft	V, ft/sec	L/D	I, sec	
6	100,000	5,950	3.6	2500	Ramjet
8	120,000	8,150	3.3	2200	Ramjet
10	133,000	10,400	3.2	1950	Scramjet
12	140,000	12,700	3.1	1700	Scramjet

TABLE 3.- RESEARCH AIRCRAFT UNFUELED WEIGHTS FOR SEVERAL TAKE-OFF WEIGHTS, CRUISE TIMES, AND CRUISE MACH NUMBERS

Take-off weight, lb	50,000				100,000				150,000			
Cruise time, sec	0	3	5	8	0	3	5	8	0	3	5	8
Cruise Mach number	Weight at end of cruise, lb											
6	28,650	28,110	27,750	27,260	57,300	56,220	55,500	54,470	85,950	84,330	83,280	81,730
8	24,000	23,480	23,130	22,610	48,000	46,940	46,230	45,220	72,000	70,420	69,360	67,820
10	20,050	20,010	19,720	19,250	40,100	39,150	38,510	37,600	60,150	58,710	57,780	56,410
12	16,700	16,290	15,980	15,600	33,400	32,540	31,990	31,140	50,100	48,800	47,970	46,740

TABLE 4.- RESEARCH AIRCRAFT WEIGHT ESTIMATES; M = 12 CRUISE FOR 5 MINUTES

	Air launch	Ground take-off	
	Advanced weight technology[a]	Advanced weight technology[a]	Ames estimate[b]
Structure (fuselage, tanks, wings, stabilizer)	11,390 (17.5)		19,330 (18.5)
Landing gear	1,020 (1.6)		3,500 (3.4)
Propulsion-rocket, J-2	2,800 (4.3)		2,800 (2.7)
Propulsion-scramjet	2,820 (4.3)		2,820 (2.7)
Subsystems	4,210 (6.4)		4,430 (4.2)
Empty weight	22,240 (34.1)	27,430 (27.4)	32,880 (31.5)
Crew and equipment	270 (0.4)		270 (0.3)
Trapped fuel	550 (0.9)		550 (0.5)
Payload	1,000 (1.5)		1,000 (1.0)
Fuel-boost and cruise LH_2	6,780 (10.4)		9,650 (9.2)
Fuel-boost lox	34,270 (52.6)		60,000 (57.4)
Fuel-descent LH_2	90 (0.1)		150 (0.1)
Gross weight	65,200 (100.0)	99,900 (100.00)	104,500 (100.0)

[a]Prepared by Messrs. D. E. Wall and H. D. Altis, McDonnell Aircraft Co., St. Louis, Mo.
[b]Prepared by Mr. L. W. Hunton, Ames Research Center.

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